NASA CR-54879
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Volume One of
Two Volumes

THERMAL PROTECTION SYSTEM FOR A CRYOGENIC SPACECRAFT PROPULSION MODULE

SUMMARY REPORT

by

W. H. Sterbentz and J. W. Baxter

prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

CONTRACT NAS 3-4199

LOCKHEED MISSILES & SPACE COMPANY / SUNNYVALE, CALIFORNIA

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FINAL REPORT

THERMAL PROTECTION SYSTEM FOR A CRYOGENIC SPACECRAFT PROPULSION MODULE

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November 15, 1966

CONTRACT NAS 3-4199

Technical Management
NASA Lewis Research Center
Cleveland, Ohio
Liquid Rocket Technology Branch
J. R. Barber and J. Kramer

LOCKHEED MISSILES & SPACE COMPANY
SUNNYVALE, CALIFORNIA 94088

FOREWORD

The Lockheed Missiles & Space Company is submitting this report in partial fulfillment of the requirements of Contract NAS 3-4199, dated June 30, 1964, "Thermal Protection System for a Cryogenic Spacecraft Propulsion Module." The report is presented in two volumes: Volume I, Summary Report; Volume 11, Technical Report. The work reported was conducted for the National Aeronautics and Space Administration through the NASA Lewis Research Center, Cleveland, Ohio. Much of the success of this Program is attributable to the timely and valuable guidance, consultation, and direction provided by Mr. J. R. Barber and J. Kramer as NASA, LeRC, Program Managers.

On the Lockheed team, the dedicated efforts of eachof the Task Leaders are both irreplaceable and invaluable to this Program. In this regard, special mention is made as follows:

Mr. B. R. Bullard – Task I and Task VI Leader

Mr. H. E. Johnson – Task II Leader

Mr. C. F. Merlet – Task III Leader

Mr. I. B. Funderburke – Task IV Leader

Mr. H. Hemesath and Mr. C. F. Merlet – Task V Leaders

Mr. J. A. Jones – Quality Assurance

Miss K. R. Grupe and Mr. R. E. Givens – Publications and Slides

Much of the functional technical data needed to make this Program an outstanding success was developed under Lockheed Independent Technology Programs, conducted prior to and concurrent with this Program. The technical contributions of the many individual contributors on these Lockheed Independent Technology Programs are also noteworthy.

The Lockheed Missiles & Space Company was indeed pleased to have conducted this Program for the NASA Lewis Research Center. The efforts of the many individual contributors of both organizations created the first successful, practical flight weight reflective multilayer insulation for liquid hydrogen tankage for long-duration space propulsion vehicles. The results of this Program and a complete description of the Thermal Protection System are presented in this report.

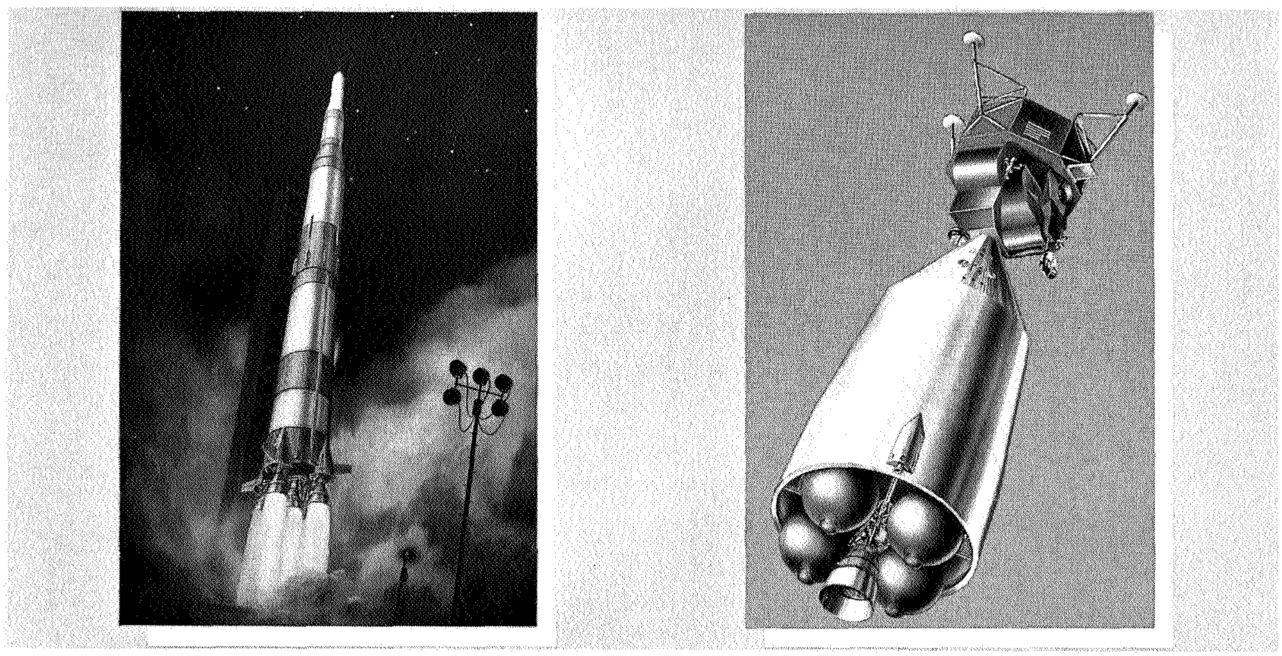
W. H. Sterbentz
J. W. Baxter
Program Managers

ILLUSTRATIONS

SUMMARY

A primary objective in the development of a thermal protection system for a cryogenic spacecraft propulsion module (Fig. 1), conducted by the Lockheed Missiles & Space Company for the National Aeronautics and Space Administration, was the development of a light-weight insulation that would limit liquid hydrogen boiloff to less than 5 percent of the initial propellant load during an 8-day lunar mission (Table 1). This program, which was both analytical and experimental in scope, answered the following additional questions posed by the NASA Lewis Research Center: (1) How can the insulation be applied to survive boost loads? (2) How can it be augmented to provide ground-hold and boost-phase insulation? (3) What is an optimum support system for the tank? In addition, the Lockheed program was aimed at developing a design based upon the practicalities of cryogenic stage manufacturing and the quality control requirements of simplicity, producibility, repeated assembly and reassembly, low cost, and controllable quality. All of these objectives were successfully achieved.

The cryogenic spacecraft propulsion module was chosen as representative of a realistic application of cryogenic propellants to a space mission. Figure 1 shows it as the



Launch Configuration | Cryogenic Propulsion Module

Fig. 1 Saturn V Launch Vehicle With Cryogenic Propulsion Module service module of a Saturn V Apollo vehicle. In this configuration, the propulsion module is 192-in. in diameter, weighs about 38,500 lb, and has a 165.2-in.-diameter liquid hydrogen tank with an impulse liquid hydrogen propellant capacity of 5500 lb. Contained in four spherical tanks nested around the RL-10 engine is the required 27,500 lb of liquid oxygen. Such a vehicle could place a lunar excursion module payload of 41,000 lb into lunar orbit, compared to the 26,000-lb payload for the current Saturn V service module using a nitrogen tetroxide/unsymmetrical-dimethyl-hydrazine (N_2O_4/UDMH) propellant combination. These marked performance gains are possible now that the objectives of this thermal protection system program have been met.

Initiated in July 1964, this development program was divided into six individual tasks:

Task I – Insulation Design
Task II – Tank Support Design
Task III – Component and Subscale Model Tests

Table 1
THE 8-DAY LUNAR MISSION PROFILE

VEHICLE PARAMETERS

Stage Total Weight (lb)	41,000
Useful Propellants (lb)	33,000
Fuel	LH_2
Oxidizer	LO_2
Stage Thrust (lb)	15,000
Tank Maximum Pressure, Pump-Fed System (psia)	25
Oxidizer/Fuel Ratio	5
Tank Venting Pressure (psia)	17
Loads:	
Acoustic	150 db (re. 0.0002 microbars), 5 to 2000 cps
Longitudinal	5-g maximum at booster burnout
Lateral	0.3-g maximum (occurs during boost when logitudinal acceleration is 2.5 g)
Dynamic	0.7-g maximum at 1 cps (wind load during ground hold)
	6-g vibratory longitudinal loads between 20 and 150 cps

PROPELLANT EXPULSION SCHEDULE

Firing	Pounds	Firing Time (sec)
1	1,000	19
2	1,000	19
3	18.000	349
4	1,000	19
5	6,600	125
6	260	5
7	260	5

SEQUENCE OF EVENTS

Event	Time (Hr After Launch)
Launch	0
(Four Earth Orbits, Each 90 Min)	
Insertion Into Lunar Trajectory	6
First Outbound Midcourse Correction	26
Second Outbound Midcourse Correction	48
Retro Into Lunar Orbit	78
(60 Hr in Lunar Orbit, Total)	
Rendezvous (Emergency)	126
Insertion Into Earth Trajectory	138
First Inbound Midcourse Correction	158
Second Inbound Midcourse Correction	186

Task IV – Half-Scale Test Model Fabrication

Task V – Half-Scale Model Tests

Task VI – Insulation Test Results Analysis and Design Modifications

In Task I, an optimum thermal protection system based on reflective multilayered insulation materials for the liquid hydrogen propellant tank of the cryogenic propulsion module was designed. Lockheed screened many different multilayered insulation system designs, narrowing the selection to three systems for more intensive evaluations from which a final selection was made. The superior insulation system (Fig. 2) was found

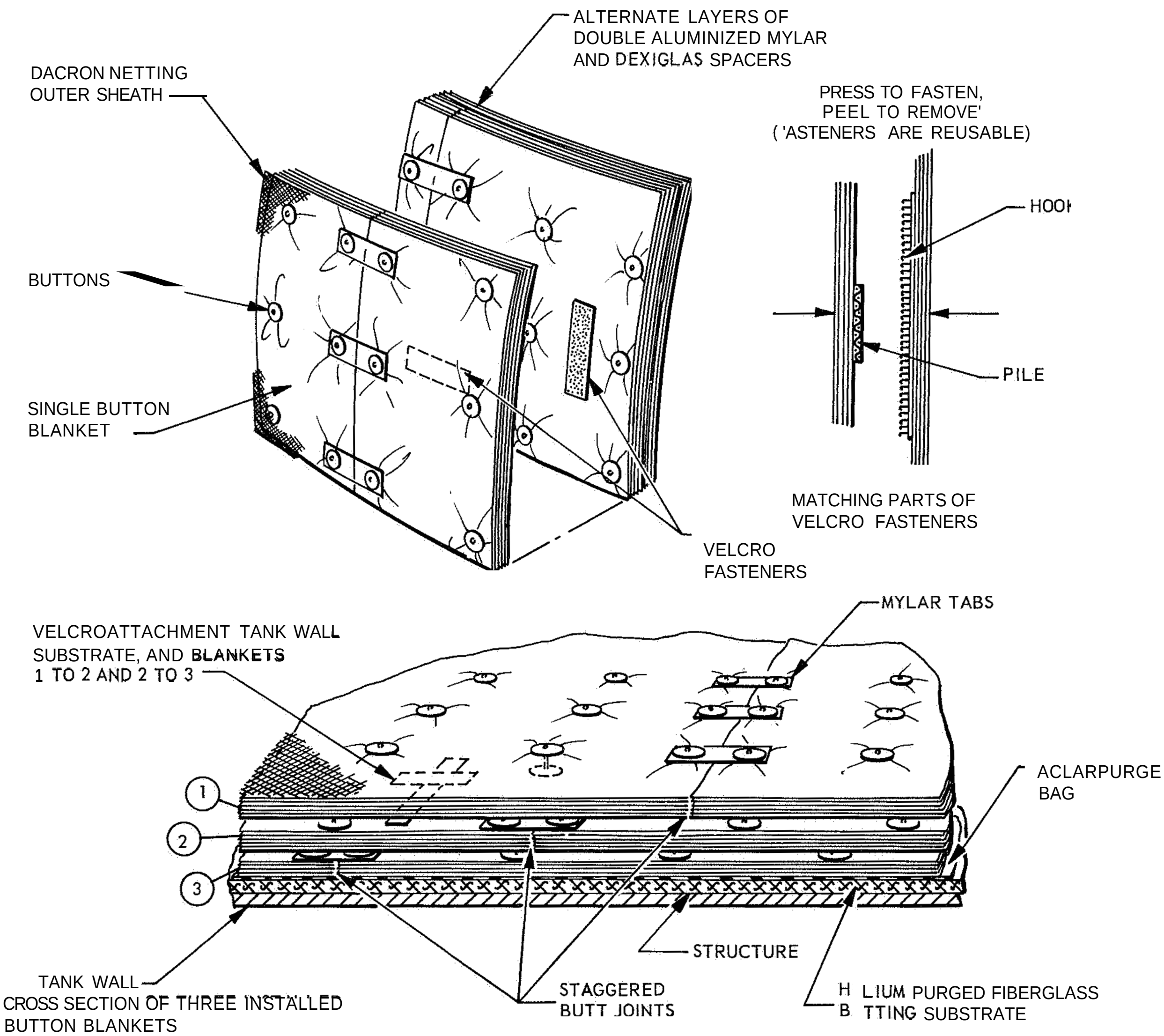


Fig. 2 Selected Insulation System for Large-Scale Tests

to be one composed of a helium-purged Fiberglas batting substrate for ground-hold insulation, overlaid with 36 layers of double aluminized 0.25-mil Mylar separated by 2.8-mil Dexiglas paper and assembled in buttoned, modular panels. The complete thermal protection system was estimated to weigh 224 lb, with a predicted 234 lb of liquid hydrogen boiloff over the entire 8-day lunar mission (Table 2). These estimates were thoroughly supported by the component and subscale tests performed with liquid hydrogen under Task III, which was conducted concurrently with Task I activities.

Also carried out concurrently with the insulation design and subscale tests was design of the support structure required to suspend the liquid hydrogen tank within the load-carrying outer shell of the vehicle. Two types of support structure – minimum-point and continuous-contact – were evaluated. Structural and minimal-thermal-conductance requirements of the tank support system were integrated in Task I insulation, design, evaluation, and selection activities. The best system for the 8-day mission was shown to be a semimonocoque continuous-contact support constructed of 0.013-in. titanium (Fig. 3).

The component and subscale tests (Task III) already mentioned were devised to prove the feasibility of the design concepts. Fundamentally, these tests fell into two broad categories: laboratory tests on cryostats (Fig. 4), calorimeters, flow conductance, and permeability apparatus; and tests conducted with 26-in.-diameter small-scale tanks (Fig. 5). Ground-hold thermal, vacuum thermal (Fig. 6), ascent rapid depressurization (Fig. 6), vibration (Fig. 7), acceleration (Fig. 8), acoustic (Fig. 7), purging, rapid thermal contraction, and combinations of these environments are

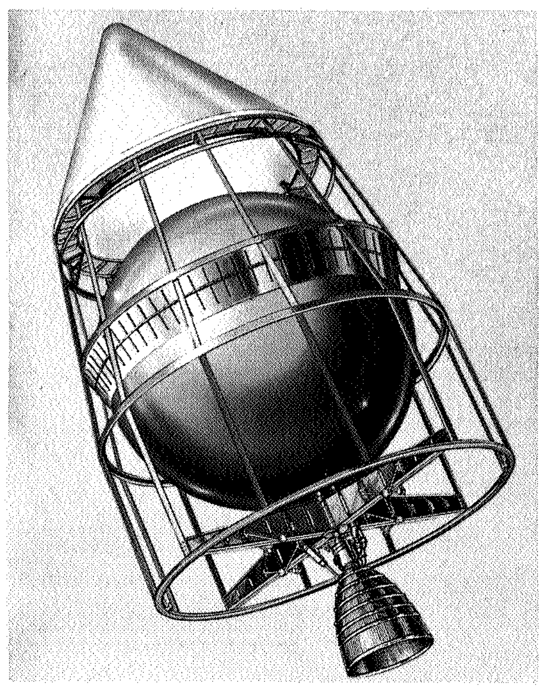


Fig. 3 Semimonocoque Titanium Support Structure

Table 2

THERMAL PROTECTION SYSTEM WEIGHT STATEMENT

Item		Weight (lb)
Tank Surface Insulation	152.8	
Velcro fasteners		1.62
Dacron		3.41
Fiberglas mat		16.85
Thread		1.15
Dacron		3.41
Purge bag		34.40
Aluminized Mylar (30 shields)		33.20
Dexiglas spacers (27 layers)		52.80
Buttons		1.78
Tape (between blankets)		0.50
Nylon net		1.66
Purge bag edges		1.24
Seams		0.67
Continuous Support Insulation	43.0	
Standoff assembly		2.73
Purge bag clamps		4.79
Standoff assembly		1.827
Purge bag tiedown assembly		3.971
Nylon net tiedown		0.005
Tank purge bag "tail"		0.717
Fiberglas mat fill		2.660
Fiberglas mat, conic section		1.740
Multilayer, conic section, outer		9.099
Multilayer, conic section, inner		8.541
Tape, Permacel PE 100		3,224
Blanket seams		0,013
Purge bag, conic section		3.70

Item		Weight (lb)
Sump Cover	9.0	
Fiberglas structure and mat		6.910
Dexiglas-Mylar blankets		1.118
Tape		0.157
Purge bag clamp		0.444
Nylon net		0.020
Stainless steel purge line		0.365
Manhole Cover	15. 5	
Fiberglas structure and mat		11.564
Dexiglas-Mylar blankets		2.618
Tape		0.196
Purge bag clamp		0.704
Nylon net		0.046
Stainless steel purge line		0.365
Plumbing Insulation	3. 8	
Fill and drain		0.714
Engine feed		0.714
Helium purge (sump)		0.373
Helium pressurization (sump)		0.373
LH_2 vent		0.714
Helium purge (manhole)		0.373
Helium pressurization (manhole)		0.548

Total System Weight: 224.1

Mission Total Liquid Hydrogen Boiloff Weight: 234

among those to which candidate insulation system were subjected. Complete structural tests of the selected insulation system were performed on a 50-deg gore section of a half-scale model of the propulsion module liquid hydrogen **tank** (Fig. 9). The design successfully passed these tests, which equalled or exceeded Saturn V launch, ascent, and space vacuum structural loads.

Following approval of the selected optimum thermal protection system by the **NASA-** LeRC Program Manager, Lockheed constructed a half-scale test model of the portion of the cryogenic spacecraft propulsion module containing the liquid hydrogen tank. The tank was fabricated of 0.125-in. -thick 2219-T87 aluminum; it had a 0.016-in. -thick monocoque titanium support cone insulated with a scale model of the selected thermal protection system. Figures 10 through 17 show the step-by-step sequence of fabricating the insulation and assembling it on the half-scale test module. This procedure allowed for easy assembly, disassembly, and reassembly of the insulation system.

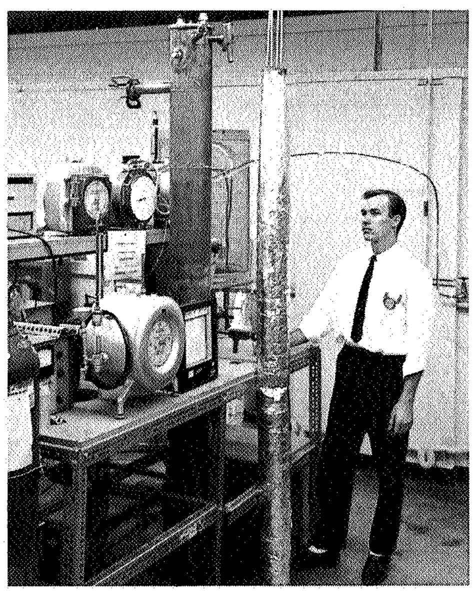


Fig. 4 Laboratory Experimentation

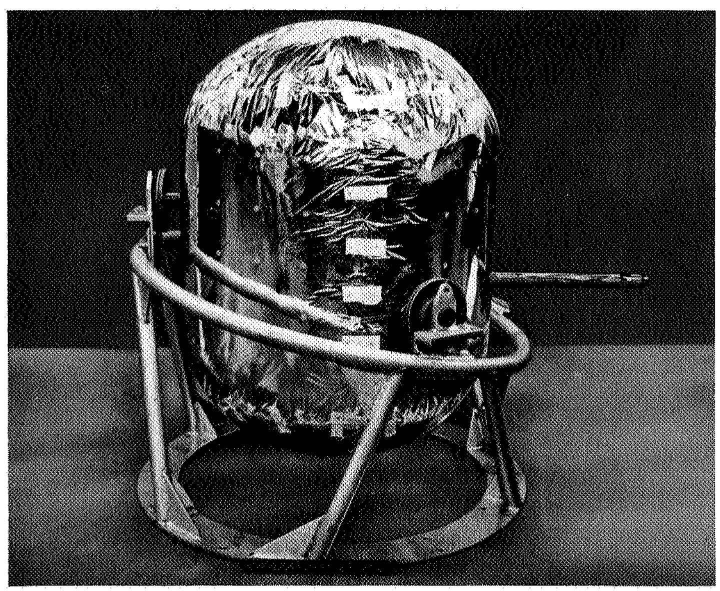


Fig. 5 Subscale Thermal Tests

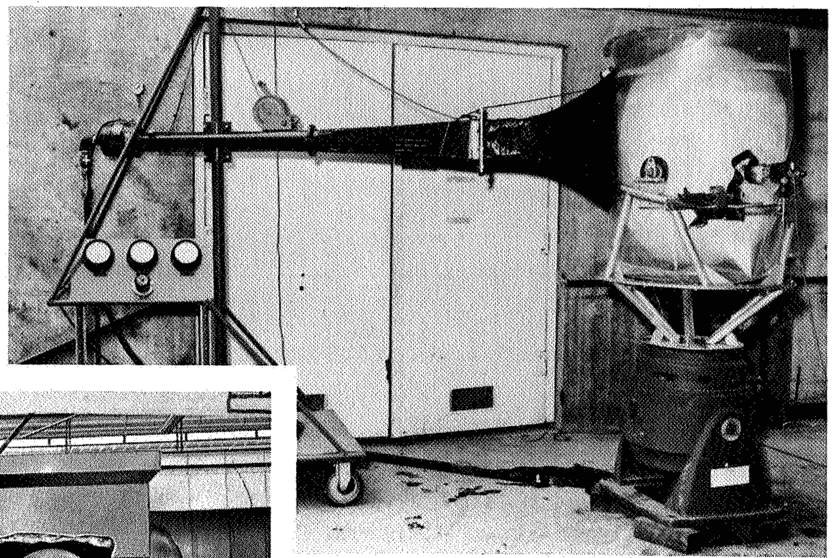


Fig. 7 Vibration and Acoustic Tests

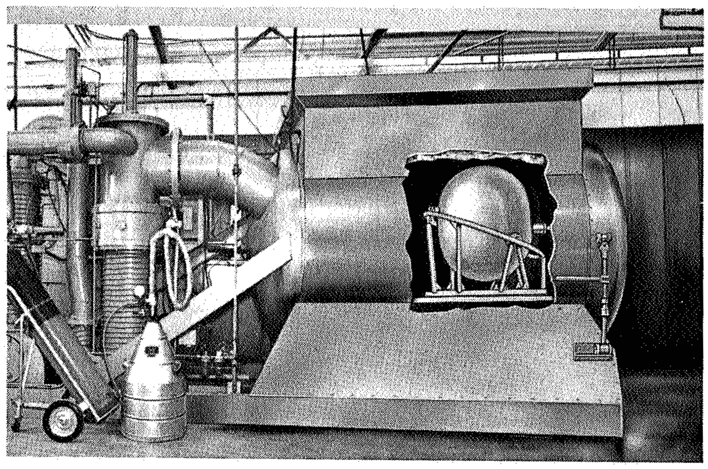


Fig. 6 Launch and Space Environment Tests

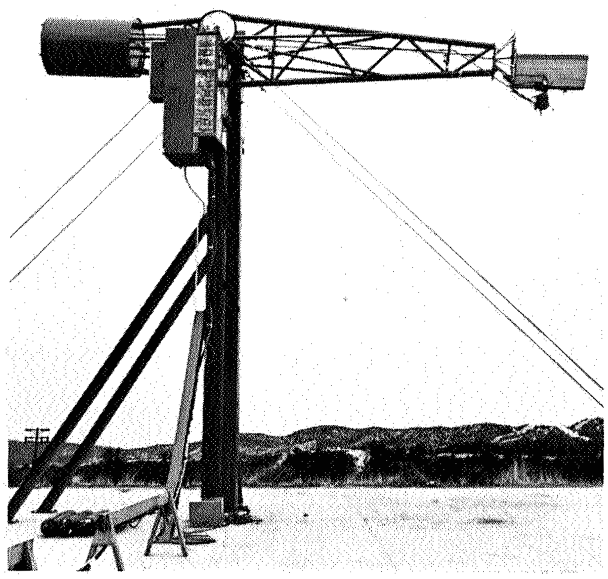


Fig. 8 Centrifuge Tests

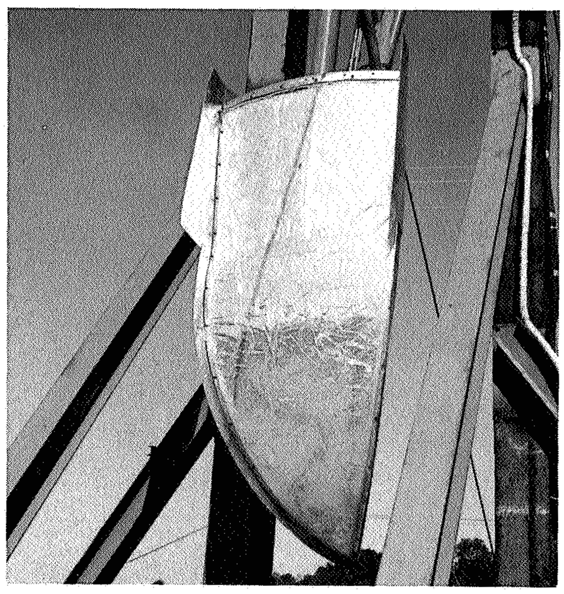


Fig. 9 Gore Panel After Final Acceleration Test

Fig. 10 Initial Layup of Aluminized Mylar and Dexiglas Paper Separators

Fig. 11 Multilayer Insulation Modules Ready for Assembly on the Liquid Hydrogen Tank

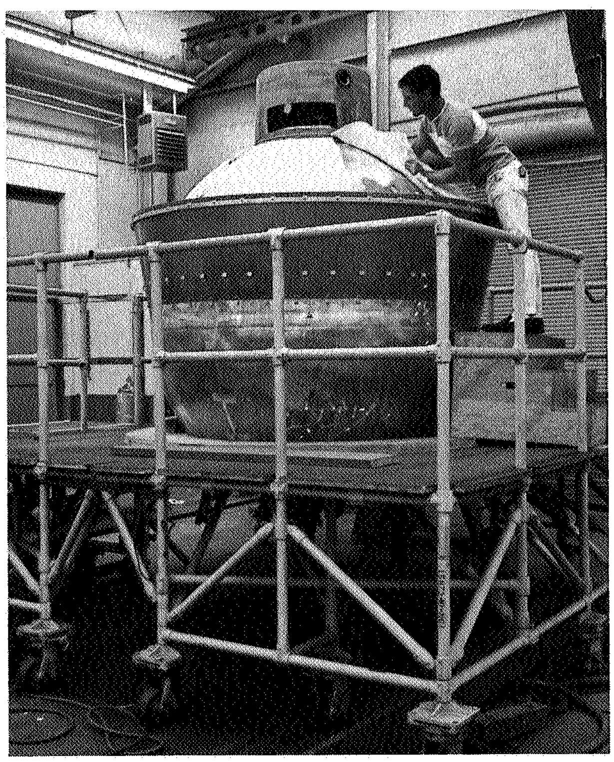


Fig. 12 Application of Fiberglas Substrate to the Tank

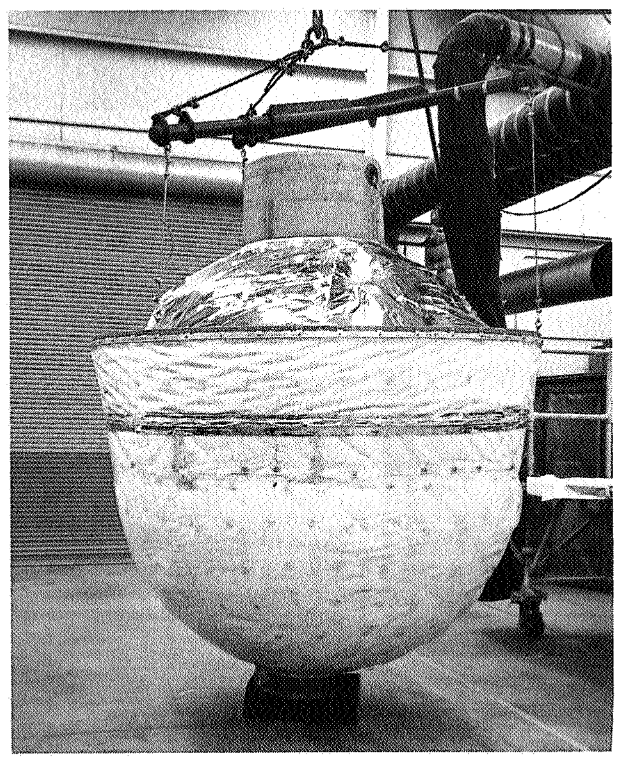


Fig. 13 Partially Assembled Insulation System Showing Both the Fiberglas Substrate and Overlaid Multilayered Aluminized Mylar

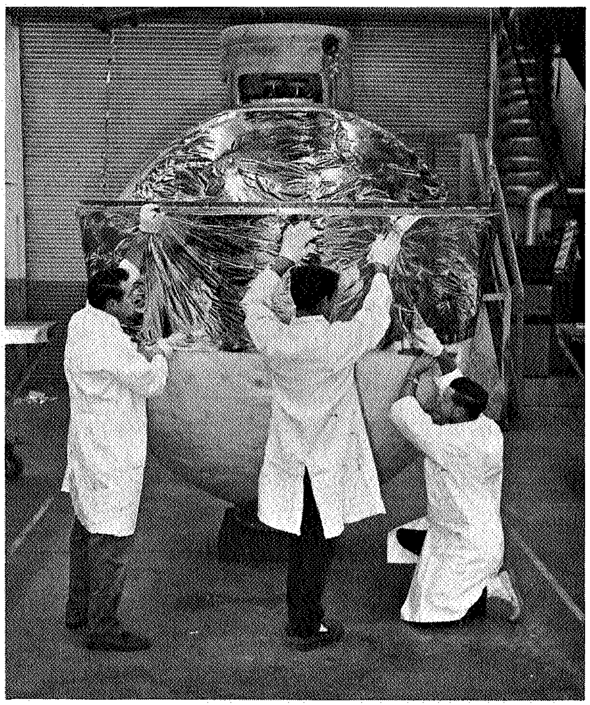


Fig. 14 Attaching Multilayer Insulation Blankets to Tank Support Cone

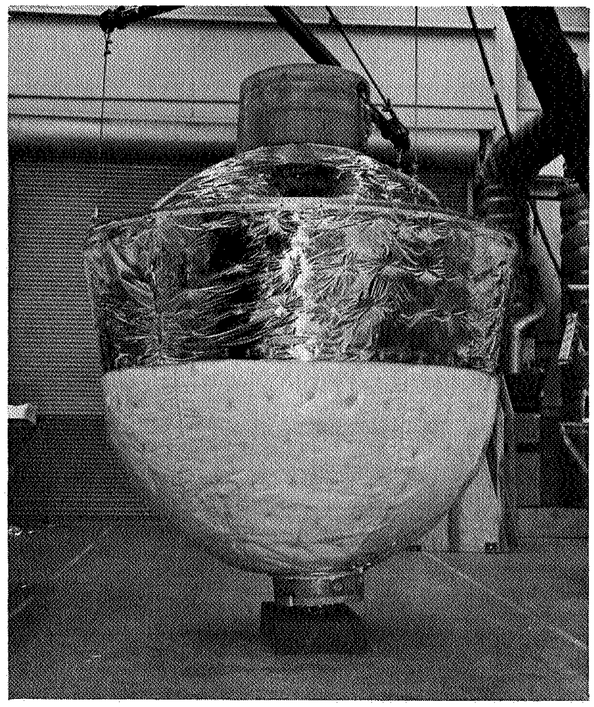


Fig. 15 Multilayer Insulation Attachment Completed on Tank Support Structure

Fig. 16 Closeup of Insulation Modules and Attachment Fasteners

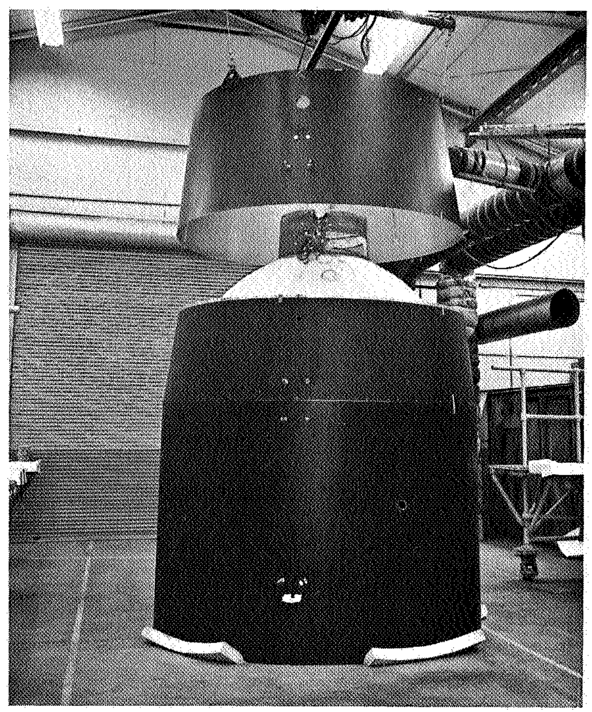


Fig. 17 Completed Insulated Liquid Hydrogen Tank in Outer Load-Carrying Shell

Having fabricated, assembled, and instrumented this half-scale model, Lockheed then tested the system under Saturn V ground-hold, ascent, and space vacuum and thermal conditions in a continuous 8-day test simulating an 8-day lunar mission. The tests were conducted in the cryogenic space vehicle flight simulator at the Lockheed Santa Cruz Test Base. This facility (Fig. 18) is 16 ft in diameter by 23 ft high; it has the capability of duplicating the Saturn V ascent pressure drop down to 1.0 psia and 10^{-6} for steady state vacuum environment, within 10 hr; ascent and space thermal heat flux is programmable by means of liquid nitrogen cold walls to simulate vehicle orientation effects. In conjunction with the comprehensive acceleration, vibration, and acoustic tests of Task 111, these tests proved the validity of the design of the selected system.

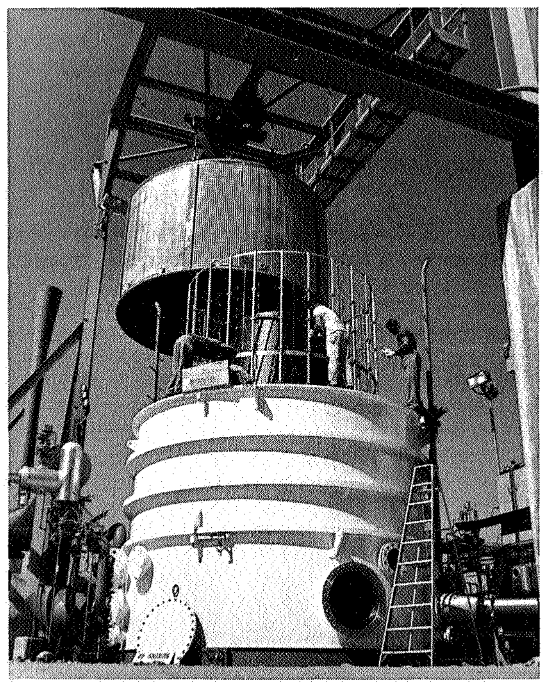


Fig. 18 Cryogenic Space Vehicle Flight Simulator With Half-Scale Propulsion Module Installed

After this successful test of the half-scale model, the insulation system was disassembled and thoroughly examined for any structural defects that may have occurred under tests.

There was no visible damage prior to disassembly. However, as the system was removed, some minor local degradations were discovered. There were marks on some of the middle blanket layers, apparently caused by friction or pressure from the top adjacent button retainer. A few buttons punctured the initial aluminized Mylar sheet only and went no further. A small number of Velcro fasteners failed, but most of these were pulled loose during disassembly. Examples of these very minor degradations are shown in Fig. 19.

The most significant damage was a tear in the insulation (Fig. 19) on the inside conical section at a laced butt joint, in the same region where buckling of the monocoque support was discovered. For economy, the tank support cone had been fabricated in pure monocoque form rather than a flight-type semimonocoque cone. It could effectively sustain only tension loads. Since the buckling of the support cone was responsible for the only significant insulation system damage and no such buckling would be expected on a flight-qualification semimonocoque structure, damage from this source is highly improbable.

Table 3 lists the original expected design performance; test results (scaled up from one-half to full size); and the modified design based upon Task V test data. As indicated,

Table 3
DESIGN AND PERFORMANCE COMPARISONS

Parameter	Configuration		
	Task I Design	Task V Results	Task VI Modified Design
No. of shields	30	30	48
Purged substrate thickness (in.)	0.4	0.4	2.0
System weight (lb)	224	224	343
Ground-hold boiloff rate (lb/hr)	230	360	36
Ascent boiloff (lb)	25	3	3
Space flight boiloff (lb)	209	443	201
Mission total boiloff (lb)	234	446	204
Boiloff to impulse propellant ratio (%)	4.3	8.1	3.7
Total weight, system and boiloff (lb)	458	671	547
Effective weight, system and 0.591 x boiloff (lb)	362	488	464

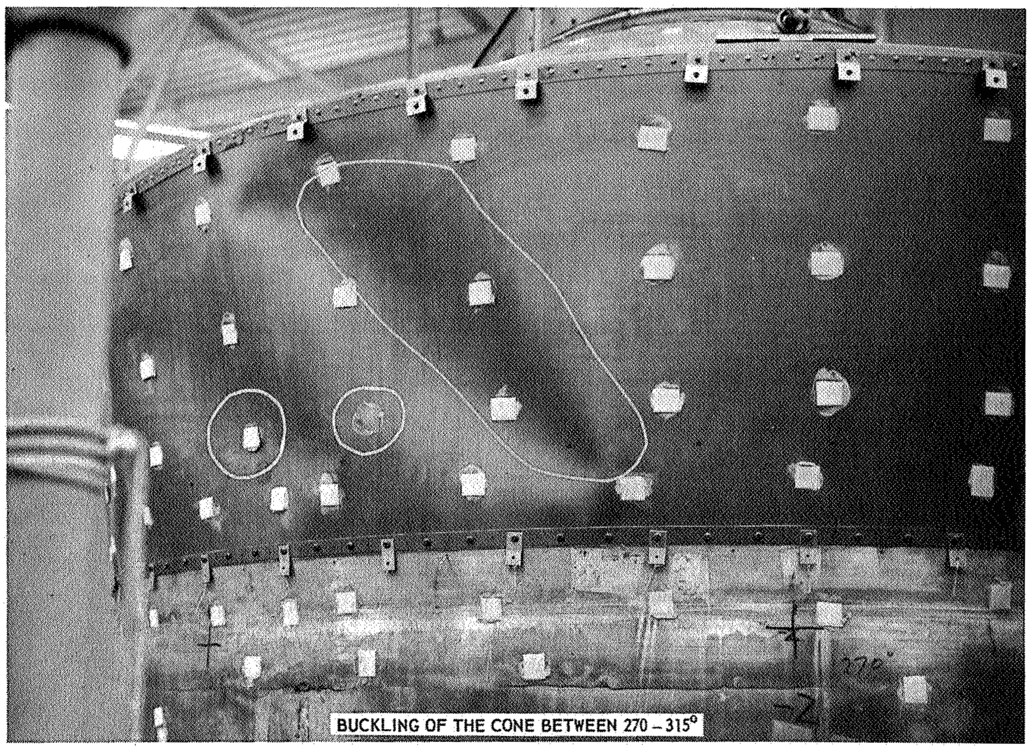
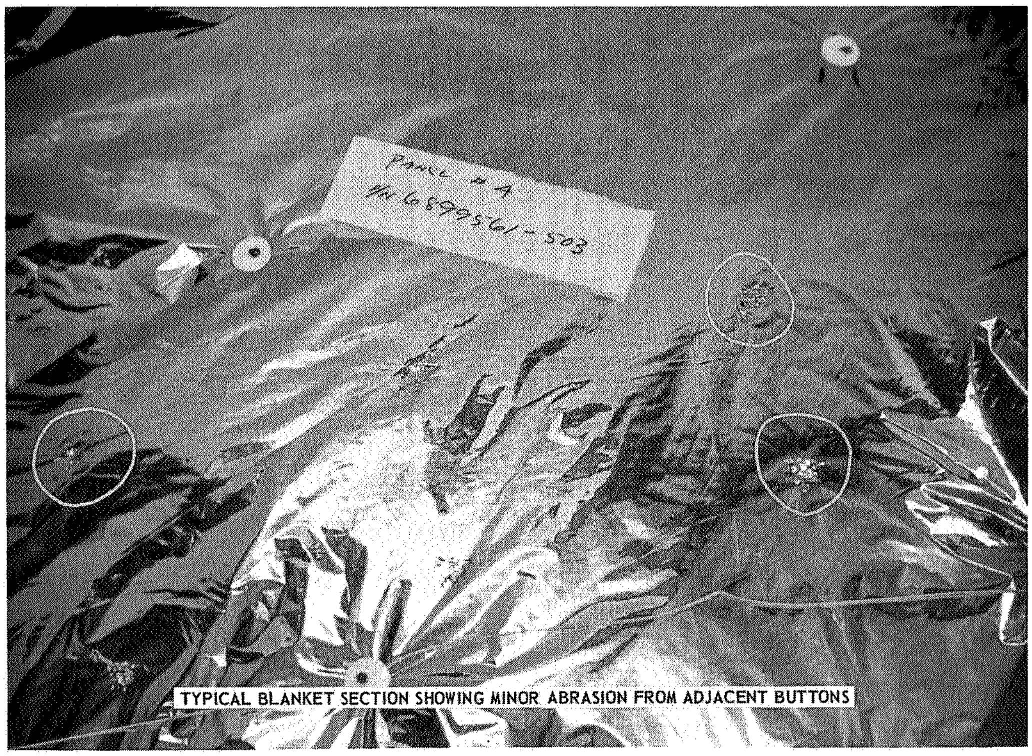
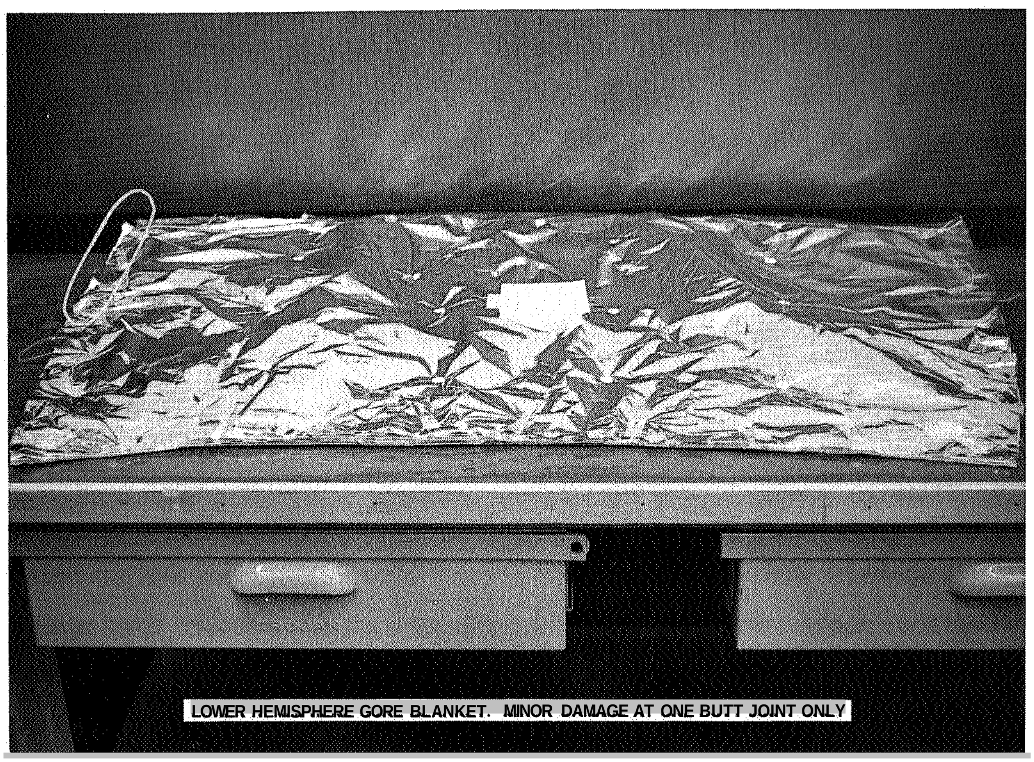


Fig. 19 Damage Sustained on the Simulated 8-Day Lunar Mission Test



TYPICAL BLANKET SECTION SHOWING MINOR ABRASION FROM ADJACENT BUTTONS



LOWER HEMISPHERE GORE BLANKET. MINOR DAMAGE AT ONE BUTT JOINT ONLY

Fig. 19 (Continued)

the total mission boiloff was 213 lb greater than expected on the basis of Task I, II, and III design and data. As a percent of mission impulse propellant, the boiloff increased from the anticipated 4.3 percent to 8.1 percent. A thorough evaluation of the sources and magnitude of heat leads into the tank experienced in Task V tests revealed certain significant facts about the Task I design features. First, a 1-in.-wide polyurethane foam thermal stop between the tank and support cone insulation blankets was both ineffective as a thermal stop and essentially transparent to thermal radiation. Second, owing primarily to inadequate insulation thickness control (1-in. to 2-1/8-in. variation) that resulted from assembly of insulation blankets constructed from flat-sheet films, the insulation heat flux was 1.25×10^{-4} Btu/sec-ft^2 compared to a predicted value of 0.83×10^{-4} Btu/sec-ft^2. Third, excellent agreement of predicted heat flux values with experiment values was obtained for all other sources of heat leak through plumbing lines and tank support structure.

Eliminating the polyurethane foam thermal stop and utilizing the experimental values of heat flux determined in Task V tests to establish the increase in insulation required to meet program objectives yielded an optimized insulation system weighing 343 lb and an estimated 3.7-percent propellant boiloff. Thus the total effective weight (insulation plus boiloff weight) of the modified system based on Task V data is estimated as 547 lb, instead of the 671 lb for the Task I design measured in Task V tests. The net weight reduction results from the boiloff reduction, which more than compensates for the increased insulation thickness and, therefore, insulation weight. Further increases in insulation weight, however, are not compensated for by equivalent decreases in boiloff.

Solving the problem of effective long-term storage of liquid hydrogen required new concepts and designs – not only of the radiation-reflective insulations and minimum heat leak tank support structure, but also of liquid hydrogen tank access hole cover and plumbing line seals, electrical and instrumentation connectors that can withstand cryogenic temperatures with no leakage of propellant or gases through them, and of the forming and welding of 2219-T87 aluminum for high-strength cryogenic tank material (Fig. 20). On this program, Lockheed demonstrated full-scale (19-1/4-in.-diameter) tank manhole cover and plumbing line seals constructed of dead soft aluminum

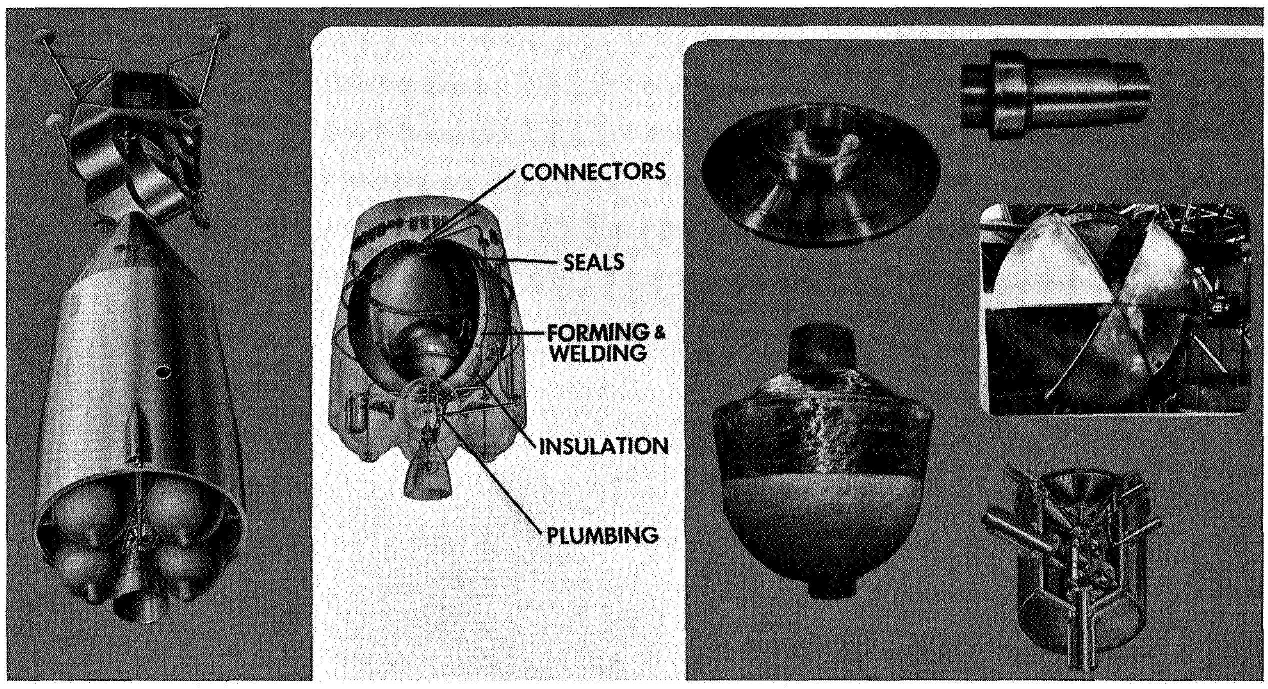


Fig. 20 New Concepts and Design Requirements for Long-Term Storage of Cryogenic Propellants

rings that minimize leakage of hydrogen or helium gas to less than 10^{-9} cm^3/sec. Explosive forming was developed to form 2219-T37 aluminum into 60-deg gore sections, which were subsequently age-sized to the -T87 condition. Total requirements for welding 2219-T87 aluminum to obtain high weld strengths were also pioneered. Consistent weld strengths exceeding 40,000 psi in 2219-T87 aluminum are now routinely achieved at Lockheed.

In summary, the large-scale ground-based technological advances demonstrated in this program have established that the potential gains available from applying cryogenic propellants to high-energy space mission can now be attained. Practical insulations and tank support structures have been developed that restrict heat flow into a liquid hydrogen tank to less than 0.35 $Btu/hr\text{-}ft^2$ from all sources, including plumbing penetrations,in a space vacuum environment. The insulation – or more correctly, the thermal protection system – has successfully withstood acceleration loads of

20 times the force of gravity; peak-to-peak amplitude vibrations up to 150 cycles/sec of 6 g; acoustic vibrations of 153 decibels; rapid depressurization at rates three times those that would be experienced in launch vehicles; ground-hold environments; and space vacuum and thermal environments. Complete details of this outstandingly successful program, including technical data and discussion of designs, procedures, and results briefly surveyed in this summary, are presented in Volume II of this report.

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